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TRANSMITTAL LETTER TO THE UNITED STATES DESIGNATED/ELECTED OFFICE (DO/EO/US) CONCERNING A FILING UNDER 35 U.S.C. 371	ATTORNEY'S DOCKET NUMBER 19391.0025 U.S. APPLICATION NO. (If known, see 37 CFR 1.5) 09/889311

INTERNATIONAL APPLICATION NO.	INTERNATIONAL FILING DATE	PRIORITY DATE CLAIM
PCT/SE00/00034	12 January 2000	18 January 1999

TITLE OF INVENTION
REDUNDANT SYSTEM FOR THE INDICATION OF HEADING AND ATTITUDE IN AN AIRCR

APPLICANT(S) FOR DO/EO/US
Peter Adebjörk, Per-Johan Nordlund, Carl-Olof Carlsson

Applicant herewith submits to the United States Designated/Elected Office (DO/EO/US) the following items and other information:

1. [x] This is a **FIRST** submission of items concerning a filing under 35 U.S.C. 371
2. ☐ This is a **SECOND** or **SUBSEQUENT** submission of items concerning a filing under 35 U.S.C. § 371.
3. [x] This express request to begin national examination procedures (35 U.S.C. 371(f)) at any time rather than delay examination until the expiration of the applicable time limit set in 35 U.S.C. 371(b) and PCT Articles 22 and 39(1).
4. [x] A proper Demand for International Preliminary Examination was made by the 19th month from the earliest claimed priority date.
5. [x] A copy of the International Application as **published** (35 U.S.C. 371(c)(2))**WO 00/42482**
   a. [x] is transmitted herewith (required only if not transmitted by the International Bureau).
   b. ☐ has been transmitted by the International Bureau.
   c. ☐ is not required, as the application was filed in the United States Receiving Office (RO/US).
6. ☐ A translation of the International Application into English (35 U.S.C. 371(c)(2)).
7. ☐ Amendments to the claims of the International Application under PCT Article 19 (35 U.S.C. 371(c)(3))
   a. ☐ are transmitted herewith (required only if not transmitted by the International Bureau).
   b. ☐ have been transmitted by the International Bureau.
   c. ☐ have not been made; however, the time limit for making such amendments has NOT expired.
   d. ☐ have not been made and will not be made.
8. ☐ A translation of the amendments to the claims under PCT Article 19 (35 U.S.C. 371(c)(3).
9. ☐ An oath or declaration of the inventor(s) (35 U.S.C. 371(c)(4)).
10. ☐ A translation of the Annexes to the International Preliminary Examination Report under PCT Article 36 (35 U.S.C. 371(c)(5)).

Items 11. to 16. Below concern other document(s) or information included:

11. [x] An Information Disclosure Statement under 37 CFR 1.97 and 1.98, PTO-1449, 4 references
12. ☐ An assignment document for recording. A separate cover sheet in compliance with 37 CFR 3.28 and 3.31 is included.
13. [x] A FIRST preliminary amendment.
    ☐ A SECOND or SUBSEQUENT preliminary amendment.
14. ☐ A substitute specification.
15. ☐ A change of power of attorney and/or address letter
16. [x] Other items or information:

PCT/ISA/210
PCT/IPEA/409

U.S. APPLICATION NO. (If known, see 37 CFR 1.5) 09/889311	INTERNATIONAL APPLICATION NO. PCT/SE00/00034	ATTORNEY'S DOCKET NUMBER 19391.0025

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X The following fees are submitted:				CALCULATIONS	PTO USE ONLY
Basic National Fee (37 CFR 1.492(a)(1)-(5)): Search Report has been prepared by the EPO or JPO........**$860.00** International preliminary examination fee paid to USPTO (37 CFR 1.482) ....**$690.00** No international preliminary examination fee paid to USPTO (37 CFR 1.482) but international search fee paid to USPTO (37 CFR 1.445(a)(2))........**$760.00** Neither international preliminary examination fee (37 CFR 1.482) nor international search fee (37 CFR 1.445(a)(2)) paid to USPTO........**$1,000.00** International preliminary examination fee paid to USPTO (37 CFR 1.482) and all claims satisfied provisions of PCT Article 33(2)-(4)........**$100.00**					
ENTER APPROPRIATE BASIC FEE AMOUNT =				**$1,000.00**	
Surcharge of **$130.00** for furnishing the oath or declaration later than ☐ 20 [x] 30 months from the earliest claimed priority date (37 CFR 1.492(e)).				**$ 130.00**	
Claims	Number Filed	Number	Rate		
Total Claims	24 - 20 =	4	X **$18.00**	**$ 72.00**	
Independent Claims	4 - 3 =	1	X **$80.00**	**$ 80.00**	
Multiple dependent claim(s)(if applicable)			+ **$270.00**	**$**	
TOTAL OF ABOVE CALCULATIONS =				**$1,282.00**	
Reduction by 1/2 for filing by small entity, if applicable.				**$**	
SUBTOTAL =				**$1,282.00**	
Processing fee of **$130.00** for furnishing the English translation later than ☐ 20 ☐ 30 months from the earliest claimed priority date (37 CFR 1.492(e)).				**$**	
TOTAL NATIONAL FEE =				**$1,282.00**	
Fee for recording the enclosed assignment (37 CFR 1.21(h)). The assignment must be accompanied by an appropriate cover sheet (37 CFR 3.28, 3.31). **$40.00** per property +				**$**	
TOTAL FEES ENCLOSED =				**$1,282.00**	
				Amount to be: Refunded	**$**
				Charged	**$1,282.00**

a. ☐ A check in the amount of $___ to cover the above fees is enclosed.

b. [x] Please charge my Deposit Account No. 19-5127; 19391.0025 in the amount of **$1,282.00** to cover the above fees. A duplicate copy of this sheet is enclosed.

c. [x] The Director is hereby authorized to charge any additional fees which may be required, or credit any overpayment to Deposit Account N 19-5127. A duplicate copy of this sheet is enclosed.

NOTE: Where an appropriate time limit under 37 CFR 1.494 or 1.495 has not been met, a petition to revive (37 CFR 1.137(a) or (b) mu be filed and granted to restore the application to pending status

SEND ALL CORRESPONDENCE TO:
Edward A. Pennington
Swidler Berlin Shereff Friedman, LLP
3000 K Street, N.W., Suite 300
Washington, DC 20007-5116

Eric Franklin
SIGNATURE

Eric J. Franklin
NAME

37,134
REGISTRATION NUMBER

09889311 .071601

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Atty Docket: 19391.0025

IN THE UNITED STATES PATENT AND TRADEMARK OFFICE

In re application of:	:	
Peter Adebjörk et al.	:	
Application No.: -- U.S. National Phase of PCT/SE00/00034	:	Examiner: --
Filed: July 16, 2001	:	Art Unit --

Title: REDUNDANT SYSTEM FOR THE INDICATION OF HEADING AND ATTITUDE IN AN AIRCRAFT

PRELIMINARY AMENDMENT

Assistant Commissioner for Patents
Washington, DC 20231

Sir:

Prior to examination, please amend the above-identification as follows:

In the Claims:

Please amend the claims as follows:

Clean Copy of Amended Claims

3. Method according to claim 1, characterised in that attitude is integrated out via information about the body-frame angular rates (p, q and r) obtained from the aircraft-fixed angular rate gyros of the aircraft.

8. Method according to claim 6, characterised in that in a second filter (22) is performed estimation of attitude errors and heading errors that arise on integration of the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros, where the estimation is done with the aid of attitude calculated from

air data information as well as derived measured body-frame magnetic field vector components.

9. Method according to claim 7, characterised in that the filtering takes place with the aid of Kalman filters.

12. Arrangement according to claim 10, characterised in that integration routine (8) integrates out the aircraft's attitude from the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros.

24. Arrangement according to claim 16, characterised in that the first filter (11) and/or the second filter (22) consists of a Kalman filter.

<u>Claim Amendments</u>

3. (Amended) Method according to claim 1 [and 2], characterised in that attitude is integrated out via information about the body-frame angular rates (p, q and r) obtained from the aircraft-fixed angular rate gyros of the aircraft.

8. (Amended) Method according to [claims 6 or 7] <u>claim 6</u>, characterised in that in a second filter (22) is performed estimation of attitude errors and heading errors that arise on integration of the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros, where the estimation is done with the aid of attitude calculated from air data information as well as derived measured body-frame magnetic field vector components.

9. (Amended) Method according to [claims 7 or 8] <u>claim 7</u>, characterised in that the filtering takes place with the aid of Kalman filters.

12. (Amended) Arrangement according to claim 10 [or 11], characterised in that integration routine (8) integrates out the aircraft's attitude from the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros.

24. (Amended) Arrangement according to [any of claims 16 - 23] claim 16, characterised in that the first filter (11) and/or the second filter (22) consists of a Kalman filter.

Remarks

Applicants have amended the claims to eliminate multiple dependencies.

Respectfully submitted,

Date: July 16, 2001

Eric J. Franklin, Reg. No. 37,134
Swidler Berlin Shereff Friedman
3000 K Street, NW, Suite 300
Washington, DC 20007
Telephone: (202) 424-7605

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09/889311
PCT/SE00/00034

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Redundant system for the indication of heading and attitude in an aircraft

TECHNICAL FIELD

The invention relates to a system function which provides display of heading and attitude on displays in an aircraft, for example a head-up display (HUD), in the event of failures in certain equipment for normal attitude display. The system function, which in English is called Attitude and Heading Reference System and is abbreviated AHRS with reference to its initials, supplements the aircraft's normal display for heading and attitude. This display is intended to help the pilot to recover from difficult attitudes and then facilitate return to base/ landing.

PRIOR ART

In order not to lose attitude and heading display in an aircraft in the event of failure of a normally-used inertial navigation system (INS) a redundant system is required. In good visibility a pilot can fly by using the horizon as an attitude reference, but with great uncertainty as to the heading. In bad weather, in cloud and at night when the horizon is not visible, the pilot can easily become disoriented and thereby place the aircraft and him/herself in hazardous situations.

AHRS systems calculate, independently of normal systems, attitude angles (pitch and roll) and heading. Such a system continuously displays the position to the pilot on a display in the cockpit. The need for a redundant system for attitude may be so great that an aircraft is not permitted to fly without one.

Redundant systems in the form of an AHRS unit are available today. Such a unit contains among other things gyros which measure aircraft angle changes in pitch, roll and yaw. It also contains accelerometers and magnetic sensor. The accelerometers are used to establish a horizontal plane. The magnetic sensors are used to obtain a magnetic north end. This type of AHRS system in the form of hardware is costly and involves the installation of heavy, bulky equipment on the aircraft. To overcome this there is proposed in this description a synthetic

AHRS which uses sensors existing in the aircraft, which are not normally intended for AHRS calculation and which therefore partly have significantly lower performance, instead of sensors of the type included in an AHRS unit.

The angles are calculated with the aid of existing sensors in the aircraft. The aim is to use existing angular rate gyro signals and support these with calculations based on other available primary data in the aircraft. Angular rate gyros are normally used in control systems and generally have substantially greater drift than gyros for navigation.

DESCRIPTION OF THE INVENTION

According to one aspect of the invention, a method is provided for synthetically calculating redundant attitude and redundant heading by means of data existing in an aircraft as specified in the claims.

Different forms of embodiment have been developed. In one embodiment the heading of the aircraft is available and in another embodiment the heading is calculated on the basis of a magnetic heading sensor. When the heading is available the calculations can be substantially reduced.

When the heading is available (redundant heading) attitude is calculated by weighting together the signals from the angular rate gyros in the flight control system of the aircraft, information from air data (altitude, speed, angle of attack) and information about heading (redundant heading).

When the heading is not available, attitude and heading are calculated according to one embodiment with the aid of Kalman filters by weighting together the signals from the angular rate gyros in the aircraft's control system, information from air data (altitude, speed, angle of attack and sideslip angle) as well as information from an existing magnetic heading detector in the aircraft.

One advantage of a synthetic AHRS according to the aspect of the invention is that it works out substantially cheaper than conventional AHRS system based on their own sensors if existing sensors in the aircraft can be used. This also saves space and weight in the aircraft.

DESCRIPTION OF FIGURES

Figure 1 shows a schematic diagram of an AHRS function in which the heading is available.

Figure 2 shows the principle for levelling of the attitude of the aircraft in a head-up display, to the left without levelling and to the right with levelling.

Figure 3 shows the block diagram of a redundant system for both attitude and heading.

Figure 4 shows in three pictures the attitude and heading of the aircraft and the axes in the body frame coordinate system, as well as the angle of attack and the sideslip angle.

Figure 5 shows how zero errors and scale factor errors impact the measured value.

DESCRIPTION OF EMBODIMENT

A number of embodiments are described below with the support of the figures. According to the invention, methods are provided for synthetically calculating attitude and heading by means of data existing in the aircraft as specified in the claims.

In a simpler embodiment, the heading of the aircraft is available. In another embodiment the heading is calculated, in this case on the basis of a magnetic heading sensor.

Calculation of AHRS when the heading is known

The signals from the three angular rate gyros 2 rigidly mounted on the body frame are used to determine the orientation of the aircraft relative to the reference coordinate system N (navigation frame). The angular rate gyros 2 measure angular velocities around the three body-frame coordinate axes (x, y, z). The angular velocities are normally designated ω_x or p (rotation around the x-axis), ω_y or q (rotation around the y-axis) and ω_z or r (rotation around the z-axis). The orientation between the body-frame coordinate system B (body) and the N

system is given by the euler angles θ, ϕ and ψ. However, since the heading is known, only θ and ϕ are of interest. With the assumption that the N system is an inertial system and is oriented so that its z-axis is parallel to the g vector of the earth, it can be shown that

$$\begin{bmatrix} \dot{\theta} \\ \dot{\phi} \end{bmatrix} = \begin{bmatrix} \omega_y \cos\phi - \omega_z \sin\phi \\ \omega_x + \tan\theta(\omega_y \sin\phi + \omega_z \cos\phi) \end{bmatrix} \tag{1}$$

If the gyros 2 were ideal, the initial values ϕ_0 and θ_0 were error-free and if the integration method used were accurate, attitude angles can be obtained by solving Eqn (1). In practice, however, none of these preconditions is satisfied; instead, sensor errors etc cause the solution to diverge and relatively soon to become unusable.

Sensor errors in the form of among others zero errors, scale factor errors, misaligned mounting and acceleration-induced drifts constitute the dominant sources of error. In level flight the zero error is the error source that dominates error growth.

Owing to sensor imperfections and uncertainty in initial values, equation (1) gives an estimate of roll and pitch angle derivatives according to

$$\dot{\hat{\varphi}} = \begin{bmatrix} \dot{\hat{\theta}} \\ \dot{\hat{\phi}} \end{bmatrix} = \begin{bmatrix} \omega_y \cos\hat{\phi} - \omega_z \sin\hat{\phi} \\ \omega_x + \tan\hat{\theta}(\omega_y \sin\hat{\phi} + \omega_z \cos\hat{\phi}) \end{bmatrix} \tag{2}$$

The difference between the expected $\hat{\varphi}_{AHRS}$ (calculated by the AHRS function) and the "actual" $\overline{\varphi}_{ref}$ (from air data, primary data calculated) attitude angles constitutes an estimate of the attitude error

$$\Delta\overline{\varphi} = \hat{\varphi}_{AHRS} - \overline{\varphi}_{ref} \tag{3}$$

See below concerning the use of $\Delta\overline{\varphi}$.

Finally the attitude angles are given as

$$\hat{\bar{\varphi}}_{AHRS} = \int_t \left(\hat{\dot{\bar{\varphi}}}\right) dt + \hat{\bar{\varphi}}_0 - lim(\Delta\bar{\varphi}) \tag{4}$$

where $\hat{\bar{\varphi}}_0$ constitutes estimated initial values.

Calculation of $\bar{\varphi}_{ref}$

The formula $\theta_{ref} = \arcsin(\dot{h}/v_t) + (\alpha * \cos\phi)$ is used when calculating θ_{ref}. $\dot{h}$ is a high-pass-filtered altitude signal. v_t is true airspeed.

The formula $\phi_{ref} = \arctan(v_t * (\dot{\psi})/g)$ is used when calculating ϕ_{ref}.

$\dot{\psi}$ is a high-pass-filtered heading (redundant heading) signal.

Zero correction of the angular rate gyros

The zero errors in the angular rate gyros 2 are heavily temperature-dependent. It may take 20 to 30 minutes for the gyros to reach operating temperature. This means that an INS failure shortly after take-off might give large zero errors if flying continued. However, it takes a certain time from gyros 2 receiving voltage to the aircraft taking off, which means that part of the temperature stabilisation has been completed when a flight begins. It is also assumed that landing can take place within a short time in the event of an INS failure during takeoff. To minimise zero errors from the angular rate gyros 2 a zero correction of the angular rate gyros is performed by software. This involves comparing the ω (p, q and r) signals from the angular rate gyros 2 with the corresponding signal from the INS, see eqn (5), by generating a difference in 4a. The difference is low-pass-filtered in a filter 5 and added to the angular rate gyro signals in a difference generator 4b, where the signal ω_k which designates the zero-error-corrected gyro signals and is used instead of ω in the AHRS calculations. This is done continuously as long as the INS is working. In the event of an INS failure the most recently performed zero corrections are used for the rest of the flight.

$$\omega_{TNS} = \begin{bmatrix} p_{TNS} \\ q_{TNS} \\ r_{TNS} \end{bmatrix} = \begin{bmatrix} \dot{\phi}_{TNS} - \dot{\psi}_{TNS}\sin\theta_{TNS} \\ \dot{\theta}_{TNS}\cos\phi_{TNS} + \dot{\psi}_{TNS}\cos\theta_{TNS}\sin\phi_{TNS} \\ -\dot{\theta}_{TNS}\sin\phi_{TNS} + \dot{\psi}_{TNS}\cos\theta_{TNS}\cos\phi_{TNS} \end{bmatrix} \tag{5}$$

A block diagram of the realisation of the AHRS function with zero correction of the angular rate gyros is shown in Figure 1. The figure gives a schematic illustration of the AHRS function. The zero correction of the angular rate gyros is performed by the units inside the dashed area D.

ψ_{TNS}, θ_{TNS} and ϕ_{TNS} are high-pass-filtered to obtain $\dot{\psi}_{TNS}$, $\dot{\theta}_{TNS}$ and $\dot{\phi}_{TNS}$. These are used in Eqn (5), which gives ω_{TNS} (p_{TNS}, q_{TNS}, r_{TNS}) in a first block 1. ω (p, q, r) which are obtained as signals from the gyros designated by 2 are low-pass-filtered in a low-pass filter 3, before the difference is generated in 4a.

The difference signal between the ω_{TNS} (p_{TNS}, q_{TNS}, r_{TNS}) signals and the ω (p, q, r) signals is low-pass-filtered with a long time-constant in a low-pass filter 5, ie its mean value is generated over a long time. The filter 5 is initialised at take-off rotation with the shorter time-constant. After a power failure, the filter 5 is initialised instantaneously.

In block 7, $\dot{\hat{\varphi}}$ is calculated, after which the integration according to Equation (4) is performed in an integrator 8, to which the initial conditions $\bar{\varphi}_0$ are added. In a difference generator 9a the signal $\Delta\bar{\varphi}$ is added, but is disconnected by means of a switch 9b under certain changeover conditions, as for example when $|\gamma| > \gamma_{LIM}$ and $|\phi| > \phi_{LIM}$. The $\Delta\bar{\varphi}$ signal passes through a limiter 9c. The magnitude of the output signal from the limiter 9c is dependent on the magnitude of the $\Delta\bar{\varphi}$ signal (ie the input signal to limiter 9c). The $\Delta\bar{\varphi}$ signal is generated according to Equation (3) in a difference generator 9d to which are added calculated $\hat{\bar{\varphi}}_{AHRS}$ attitude angles and "actual" $\bar{\varphi}_{ref}$ attitude angles from sensors (primary data) designated with 9e.

Despite compensations, the calculated angles from the AHRS contain minor zero errors. Since the output signals are used for head-up display, this is corrected by using $\Delta\phi$ in roll and $\Delta\theta$ in

pitch to level the SI image until a stable position is obtained. See Figure 2, where the line H symbolises the actual horizon and where an aircraft is represented by P. Note that this levelling of the HUD only takes place when one is within the limits described above.

AHRS calculation when the heading is also to be calculated

Figure 3 shows schematically the modules that form building blocks for another variant of a synthetic AHRS and how these modules are linked together to create a redundant attitude and a redundant heading.

Figure 3 shows the principle of the redundant system in accordance with the aspect of the invention. The system consists of two subsystems A and B; the first subsystem A performs estimation of any errors in the measured geomagnetic field and the other subsystem B performs calculation of redundant attitude and heading. In all, this results in five building blocks, where a first measurement routine 10 and a first Kalman filter 11 constitute the building blocks in the first subsystem A and further where the integration routine (1/s) 20, measurement routine 21 and a second Kalman filter 22 constitute the building blocks in the second subsystem B.

With measurement routine 10, measured field vector components in the body frame coordinate system are transformed, to a north-, east- and vertically-oriented coordinate system called the navigation frame. The transformation takes place with the aid of attitude and heading from the inertial navigation system of the aircraft, INS, via wire 12. The field vector components of the geomagnetic field are taken from a magnetic heading sensor in the aircraft and arrive via wire 13. In the first Kalman filter 11, the errors in the field vector components are then estimated on the basis of knowledge about the nominal nature of the components, after which the estimated values are stored in a memory 14.

Subsystem A (measurement routine 10 and Kalman filter 11) are used only when the INS is working correctly. In the event of INS failure, the latest possible estimate of the errors in the field vector components is used, ie that which has been stored in memory 14. Since it may be difficult in many cases to decide whether the INS is working as it should, the absolutely last estimate should not be used. In order to solve this, the estimates of errors in the measured

geomagnetic field that are used are at least one flight old. ie the estimates that are stored in the memory from the previous flight or earlier.

The integration routine 20 receives information about angular velocities, in this case for the three coordinate axes x, y and z in the body frame. These are normally designated ω_x or p (rotation around the x-axis), ω_y or q (rotation around the y-axis) and ω_z or r (rotation around the z-axis). The information is taken from the angular rate gyros of the control system and is fed via wire 15 to routine 20 which integrates out attitude and heading via a transformation matrix.

The second measurement routine 21 consists of a developed variant of the first measurement routine 11 and uses the field vector components derived from the first measurement routine 11. In addition, a roll and pitch angle are calculated with the aid of data from existing air data and existing slip sensors, data which arrives to measurement routine 21 via wire 16 to measurement routine 21. By means of the second Kalman filter 22 the attitude and heading errors that arise on integration of the angular rate gyro signals of the control system are primarily calculated. Secondarily, Kalman filter 22 is used to estimate the biases in the angular rate gyros, ie the biases in p, q, and r.

The first measurement routine 10

The geomagnetic field can be calculated theoretically all over the world. To do this, the IGRF (International Geomagnetic Reference Field) is used, for example.
The field vector in the body frame is designated here with B_B and the field vector in the navigation frame with B_N. Further, the three components of the field vector are designated in accordance with

$$B = [B_x, B_y, B_z]^T . \tag{6}$$

With the aid of the transformation matrix C_B^N, which transforms a vector from body frame to navigation frame, we have

$$B_N = C_B^N \cdot B_B , \tag{7}$$

where C_B^N has the appearance

$$C_B^N = \begin{bmatrix} c_{11} & c_{12} & c_{13} \\ c_{21} & c_{22} & c_{23} \\ c_{31} & c_{32} & c_{33} \end{bmatrix} \tag{8}$$

The transformation matrix C_B^N is calculated with the aid of attitude and heading, θ, ϕ, ψ, from the INS.

The difference between a measured field vector and a field vector calculated in accordance with the model will be

$$B_{N,\text{ measured}} - B_{N,\text{ calculated}} = C_B^N \cdot \delta B_B$$

where δ designates the difference between the measured and the calculated quantity.

The left-hand part of Eqn (9) becomes the output signal from the first measurement routine 10 and thus the input signal to Kalman filter 11. Further, the right-hand part of Eqn (9) is used in Kalman filter 11, which is evident from the description of the first Kalman filter 11 below.

The first Kalman filter 11

Given the state model

$$\begin{aligned} x_{k+1} &= F_k x_k + w_k \\ z_k &= H_k x_k + e_k, \end{aligned} \tag{10}$$

a Kalman filter works in accordance with:

Time updating

$$\begin{aligned} x^-_{k+1} &= F_k x^+_k \\ P^-_{k+1} &= F_k P^+_k F^T_k + Q_k, \end{aligned} \tag{11}$$

where P^-_{k+1} is the estimated uncertainty of the states after time updating.

Measurement updating

$$\begin{aligned}
K_{k+1} &= P^-_{k+1}H^T_{k+1}[H_{k+1}P^-_{k+1}H^T_{k+1}+R_{k+1}]^{-1} \\
\hat{x}^+_{k+1} &= \hat{x}^-_{k+1}+K_{k+1}[z_{k+1}-H_{k+1}\hat{x}^-_{k+1}] \\
P^+_{k+1} &= P^-_{k+1}-K_{k+1}H_{k+1}P^-_{k+1},
\end{aligned} \tag{12}$$

where P^+_{k+1} is the estimated uncertainty of the states after measurement updating.

The errors in the field vector components are modelled according to

$$\begin{bmatrix}\delta B_x\\ \delta B_y\\ \delta B_z\end{bmatrix} = \begin{bmatrix}b_x\\ b_y\\ b_z\end{bmatrix} + \begin{bmatrix}s_x & k_{xy} & k_{xz}\\ k_{yx} & s_y & k_{yz}\\ k_{zx} & k_{zy} & s_z\end{bmatrix}\cdot\begin{bmatrix}B_x\\ B_y\\ B_z\end{bmatrix}, \tag{13}$$

where b are biases, s are scale factor errors and k is a cross-coupling from one component to another (for example, index xy refers to how the y-component affects the x-component). These 12 errors can represent the states in the first Kalman filter 11 according to

$$x_k = \begin{bmatrix}b_x\ b_y\ b_z\ s_x\ s_y\ s_z\ k_{xy}\ k_{xz}\ k_{yz}\ k_{yx}\ k_{zx}\ k_{zy}\end{bmatrix}^T \tag{14}$$

and each of the state equations looks like this

$$x_{k+1} = x_k + w_k, \tag{15}$$

where the index k designates the time-discrete count-up in time.

In Eqn (15), w_k is a weakly time-discrete process noise to model a certain drift in the errors. Eqn (15) means that the prediction matrix becomes the unit matrix and the covariance matrix for the process noise will be the unit matrix multiplied by σ^2_w, where σ_w is typically set to one hundred-thousandth (dimensionless since the field vector components are normalised to the amount 1 before they are used).

Where measurement updating of Kalman filter 11 is concerned, Eqn (9) is used and the measurement matrix looks like this

$$H_k = \begin{bmatrix}c_{11} & c_{12} & c_{13} & c_{11}B_x & c_{12}B_y & c_{13}B_z & c_{11}B_y & c_{11}B_z & c_{12}B_z & c_{12}B_x & c_{13}B_x & c_{13}B_y\\ c_{21} & c_{22} & c_{23} & c_{21}B_x & c_{22}B_y & c_{23}B_z & c_{21}B_y & c_{21}B_z & c_{22}B_z & c_{22}B_x & c_{23}B_x & c_{23}B_y\\ c_{31} & c_{32} & c_{33} & c_{31}B_x & c_{32}B_y & c_{33}B_z & c_{31}B_y & c_{31}B_z & c_{32}B_z & c_{32}B_x & c_{33}B_x & c_{33}B_y\end{bmatrix} \tag{16}$$

Owing to unmodelled interference, the measured geomagnetic field vector will deviate from the model, both in direction and in amount. The simplest variant is to model this interference as a constant white measurement noise with the aid of the measurement noise covariance matrix R_k. The standard deviations for the measurement noise for the three field vector component measurements are each typically set to one-tenth (dimensionless because the field vector components are normalised to 1 before they are used).

A Chi2 test is used to avoid the impact of bad measurements. In addition, the measurements of the field vector components are not used if the angular velocities are too high. The reason for this is that various time delays exert an effect at high angular velocities.

Integration routine 20

It can be shown that the time-derivative of the transformation matrix C_B^N becomes

$$\dot{C}_B^N = C_B^N \cdot W_{IB} - W_{IN} \cdot C_B^N . \tag{17}$$

In Eqn (17) W_{IB} and W_{IN} are, respectively, B's (body frame) rotation relative to I (inertial frame) and N's (navigation frame) rotation relative to I. Both are written in matrix form. Since we are concerned with redundant attitude and redundant heading, where the requirements for attitude errors are of the order of 2 degrees, whilst the elements in W_{IN} are of the order of 0.01 degrees, W_{IN} is disregarded. The expression in (17) will then be

$$\dot{C}_B^N = C_B^N \cdot W_{IB} , \tag{18}$$

where W_{IB} is the angular rate gyro signals from the angular rate gyros of the control system. In principle Eqn (18) means that there are nine differential equations. Because of orthogonality, only six of these need be integrated and the other three can be calculated with the aid of the cross-product.

The second measurement routine 21

The second measurement routine 21 consists of a developed variant of the first measurement routine 11, in which the expansion consists of calculating the roll and pitch angles with the aid of data from air data (altitude and speed) and the slip sensors (angle of attack and sideslip angle).

In the first measurement routine 11 it is assumed that only the field vector components are incorrect and that attitude and heading are correct. This assumption is reasonable because the field vector components are resolved with the aid of attitude and heading from the INS. In the second measurement routine 21 this is not satisfied, and consideration must also be given to attitude and heading errors. The field vector used in the second measurement routine is compensated for errors estimated in subsystem A.

Errors in both the field vector and the transformation matrix mean that

$$B_{N,\,Measured} = \hat{C}_B^N \cdot B_{B,\,Measured} \tag{19}$$

where $\hat{C}_B^N$ stands for calculated transformation matrix and means that

$$\hat{C}_B^N = C_B^N + \delta C_B^N . \tag{20}$$

If we use (20), generate the difference between measured and calculated field vector and disregard error products, we get

$$B_{N,\,Measured} - B_{N,\,\text{Calculated}} \approx \delta C_B^N \cdot B_{N,\,\text{Measured}} + \hat{C}_B^N \cdot \delta B_B \; . \tag{21}$$

In the second measurement routine 21, roll and pitch angle are calculated with the aid of altitude, speed, angle of attack and sideslip angle. The pitch angle can be calculated according to

$$\theta_{\text{ref}} = \operatorname{asin}\left(\frac{\dot{h}}{v}\right) + \cos(\phi)\alpha + \sin(\phi)\beta \tag{22}$$

To be able to calculate the pitch angle according to the expression in Eqn (22) an altitude derivative is required. This altitude derivative is not directly accessible and must instead be calculated on the basis of existing altitude which is obtained from air data. The calculation is done according to

$$\dot{h} = \dot{h}(n) = \frac{1}{\tau}\left(\left(\tau - \frac{1}{f_s}\right) \cdot \dot{h}(n-1) + h(n) - h(n-1)\right), \tag{23}$$

ie a high-pass filtering of the altitude. The symbols τ and f_s in Eqn (23) represent respectively the time-constant of the filtering and the sampling frequency. The speed v used in Eqn (22) is approximately v_t (true speed relative to the air). By approximately we mean that, when calculating v_t, measured temperature is not used, which is the normal case, but a so-called standard temperature distribution is used here instead.

Further, the roll angle can be calculated according to

$$\phi_{\mathrm{ref}} = \operatorname{atan}\frac{v\dot{\psi}}{g}. \tag{24}$$

The expression in Eqn (24) applies only for small roll and pitch angles, small angular velocities and moreover when the angles of attack and the sideslip angles are small.

The above two expressions are calculated and compared with the attitude that is calculated via the integration routine by generating the difference according to

$$\begin{aligned} \phi - \phi_{\mathrm{ref}} &= \operatorname{atan}\frac{c_{32}}{c_{33}} - \operatorname{atan}\frac{v(c_{33} \cdot \omega_z + c_{32} \cdot \omega_y)}{g(c_{11}^2 + c_{21}^2)} \\ \theta - \theta_{\mathrm{ref}} &= \operatorname{atan}\frac{-c_{31}}{\sqrt{1 - c_{31}^2}} - \left(\operatorname{asin}\left(\frac{\dot{h}}{v}\right) + \cos\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\alpha + \sin\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\beta\right), \end{aligned} \tag{25}$$

where

$$\begin{aligned} \phi &= \operatorname{atan}\frac{c_{32}}{c_{33}} \\ \theta &= \operatorname{atan}\frac{-c_{31}}{\sqrt{1 - c_{31}^2}} \\ \dot{\psi} &= \frac{c_{33} \cdot \omega_z + c_{32} \cdot \omega_y}{c_{11}^2 + c_{21}^2}. \end{aligned} \tag{26}$$

The second Kalman filter 22

The second Kalman filter 22 can be said to be the heart of the system. Here are estimated the attitude and heading errors that arise on integration of the angular rate gyro signals from the flight control system. Also estimated are the zero errors in the field vector components of the angular rate gyro signals. Further, possible residual errors in the field vector components, ie the errors that the first Kalman filter 11 cannot reach are estimated here. All in all, this means nine states: three for attitude and heading errors, three for the zero errors in the angular rate gyro signals and three for residual errors in the field vector components (three zero errors). Attitude and heading errors are represented by a rotation of the body-frame system from a calculated to a true coordinate system. The error in $\hat{C}_B^N$ can be written

$$\delta C_B^N = \hat{C}_B^N - C_B^N = C_B^N \cdot C_{\hat{B}}^B - C_B^N = C_B^N \cdot (C_{\hat{B}}^B - I) . \tag{27}$$

One can ascertain that

$$C_{\hat{B}}^B = \begin{bmatrix} 1 & -\gamma_z & \gamma_y \\ \gamma_z & 1 & -\gamma_x \\ -\gamma_y & \gamma_x & 1 \end{bmatrix} = \Gamma + I , \tag{28}$$

where Γ is the matrix form of $\gamma = [\gamma_x, \gamma_y, \gamma_z]^T$ and I is the unit matrix (T means transponate).

The elements of the vector γ describe a small rotation around the respective axis between actual (true) and calculated body frame system. The corresponding differential equations for the elements of γ can be derived to

$$\dot{\gamma} = \delta\omega , \tag{29}$$

where $\delta\omega$ is the errors in the angular rates from the angular rate gyros.

The errors in the angular rates are modelled as three first-order Markov processes according to

$$\delta\dot{\omega} = -\frac{1}{\tau_\omega}\delta\omega + u_\omega \tag{30}$$

where the time-constant τ_ω is set typically to a number of hours and the three u_ω to typically less than one degree/second.

Residual errors in the field vector components are modelled (the zero errors) in a similar way, ie

$$\dot{b} = -\frac{1}{\tau_b}b + u_b \tag{31}$$

where τ_b is set typically to a number of hours, and u_b is set typically to a few hundredths (dimensionless because the field vector components are normalised to 1 before they are used).

This gives a state vector according to

$$x_k = \begin{bmatrix}\gamma_x \ \gamma_y \ \gamma_z \ \delta\omega_x \ \delta\omega_y \ \delta\omega_z \ b_x \ b_y \ b_z\end{bmatrix}^T \tag{32}$$

and a prediction matrix according to

$$F_k = I + \int_t^{t+\Delta T} A(\tau)d\tau\,, \tag{33}$$

where $A(\tau)$ is the matrix that described the time-continuous state equations as above. The covariance matrix for the process noise Q_k is set to a diagonal matrix. Among other things, u_ω and u_b described above are used as diagonal elements. As regards the diagonal elements linked to the states for attitude and heading errors (the first three), the effects of the scale factor errors in the angular rate gyros are included. These scale factor errors are normally of the order of 2% and can cause major errors in integrated-out attitude and heading at high angular rates.

The measurements are five in number: three derived field vector components and roll and pitch angle calculated from air data. These measurements are obtained by using the relations (21) and (25).

As regards the measurement matrix H_k, relation (21) is used to fill out the three top lines. This results in the three top lines of the matrix having the appearance

$$H_{k,1-3} =$$

$$\begin{bmatrix} c_{13}B_y - c_{12}B_z & c_{11}B_z - c_{13}B_x & c_{12}B_x - c_{11}B_y & 0 & 0 & 0 & c_{11} & c_{12} & c_{13} \\ c_{23}B_y - c_{22}B_z & c_{21}B_z - c_{23}B_x & c_{22}B_x - c_{21}B_y & 0 & 0 & 0 & c_{21} & c_{22} & c_{23} \\ c_{33}B_y - c_{32}B_z & c_{31}B_z - c_{33}B_x & c_{32}B_x - c_{31}B_y & 0 & 0 & 0 & c_{31} & c_{32} & c_{33} \end{bmatrix} \tag{34}$$

For the last two lines of H_k Eqn (25) is used, by differentiating the two right-hand parts with respect to all states in the second Kalman filter 22. As a result, the last two lines get the elements (the index designates row and column in that order)

$$
\begin{aligned}
h_{41} &= 1-\frac{vg(c_{33}\omega_y-c_{32}\omega_z)(c_{11}^2+c_{21}^2)}{g^2(c_{11}^2+c_{21}^2)^2+v^2(c_{33}\cdot\omega_z+c_{32}\cdot\omega_y)^2}\\
h_{42} &= \frac{2vg(-c_{11}c_{13}-c_{21}c_{23})(c_{33}\omega_z+c_{32}\omega_y)-vg\omega_z c_{31}(c_{11}^2+c_{21}^2)}{g^2(c_{11}^2+c_{21}^2)^2+v^2(c_{33}\cdot\omega_z+c_{32}\cdot\omega_y)^2}-\frac{c_{31}c_{32}}{c_{32}^2+c_{33}^2}\\
h_{43} &= \frac{2vg(c_{11}c_{12}+c_{21}c_{22})(c_{33}\omega_z+c_{32}\omega_y)+vg\omega_y c_{31}(c_{11}^2+c_{21}^2)}{g^2(c_{11}^2+c_{21}^2)^2+v^2(c_{33}\cdot\omega_z+c_{32}\cdot\omega_y)^2}-\frac{c_{31}c_{33}}{c_{32}^2+c_{33}^2}\\
h_{45} &= \frac{vgc_{32}(c_{11}^2+c_{21}^2)}{g^2(c_{11}^2+c_{21}^2)^2+v^2(c_{33}\cdot\omega_z+c_{32}\cdot\omega_y)^2}\\
h_{46} &= -\frac{vgc_{33}(c_{11}^2+c_{21}^2)}{g^2(c_{11}^2+c_{21}^2)^2+v^2(c_{33}\cdot\omega_z+c_{32}\cdot\omega_y)^2}
\end{aligned}
\tag{35}
$$

and

$$
\begin{aligned}
h_{51} &= \sin\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\alpha-\cos\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\beta\\
h_{52} &= \frac{c_{33}}{\sqrt{1-c_{31}^2}}+\frac{c_{32}c_{31}}{c_{32}^2+c_{33}^2}\left(-\sin\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\alpha+\cos\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\beta\right)\\
h_{53} &= -\frac{c_{32}}{\sqrt{1-c_{31}^2}}-\frac{c_{33}c_{31}}{c_{32}^2+c_{33}^2}\left(\sin\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\alpha-\cos\left(\operatorname{atan}\frac{c_{32}}{c_{33}}\right)\beta\right).
\end{aligned}
\tag{36}
$$

The remaining elements in the fourth and fifth line are zero.

The simplest choice for the covariance matrix for the measurement noise R_k is a diagonal matrix. The first four measurement noise elements have a standard deviation which is set typically to one-tenth. The fifth measurement noise element on the other hand has a standard deviation that is set to a function of the altitude derivative and the speed. The function is quite simply a scaled sum of the expression for calculating pitch angle and according to Eqn (25) differentiated with respect to the altitude derivative and the speed. The function is set to

$$
f(\dot{h}, v) = 5\cdot\left|\frac{\partial\theta_{\text{ref}}}{\partial\dot{h}}\right|+50\cdot\left|\frac{\partial\theta_{\text{ref}}}{\partial v}\right| \tag{37}
$$

and gives a measure of the sensitivity of the pitch angle calculation to errors in the altitude derivative and the speed.

Since the errors in attitude and heading calculated with the aid of the integration routine grow rapidly, estimated attitude and heading errors must be fed back to the integration routine,

which is done with wire 17. If this is not done, the error equations in the second Kalman filter 22 rapidly become invalid by reason of the fact that the equations are fundamentally non-linear. In addition, the estimates of the zero errors in the angular rate gyros are fed back via a wire 18. This results in better linearisation of the second Kalman filter 22 and furthermore the sampling frequency f_s can be kept down.

In some flying situations the calculations that are performed in the second measurement routine 21 are inferior, either because the measurement equations are not sufficiently matched or because the measurement data is inherently poor. Calculation of the roll angle from air data is used only in level flight. No measurement is used if the angular rates are not sufficiently small, typically a couple of degrees or so per second. The measurement residuals are also checked, where the measurement residuals are not allowed to exceed typically one to two times the associated estimated uncertainty.

Symbols

Coordinate systems

I (Inertial frame): a system fixed in inertial space.
When flying above the surface of the earth it is customary for the centre of this system to coincide with the centre of the earth. This is really an approximation, since a system fixed in inertial space must not rotate. Because the earth rotates around the sun, the I-system will also rotate. However, the error that arises is negligible. The accelerations and angular rates measured by the sensors in an inertial navigation system are relative to that system.
N (Navigation frame): a system with its centre in the aircraft and with its xy plane always parallel to the surface of the earth.
The x-axis points to the north, the y-axis to the east and the z-axis vertically down towards the surface of the earth.
B (Body frame): a system in the aircraft, fixed to the body frame.
This coordinate system rotates with the aircraft. The x-axis points out through the nose, the y-axis through the starboard wing and the z-axis vertically down relative to the aircraft.

Table 1 Explanation of designations (symbols) for angles and angular rates. See also Figure 4.

ϕ	Angle between y_B and the horizontal plane, tilted by the angle θ along x_B (roll angle).
ϕ_0, $\hat{\phi}$	Initial value for the roll angle and estimated roll angle, respectively
ϕ_{ref}	Roll angle calculated with the aid of data from air data and the heading derivative
θ	Angle between x_B and the horizontal plane (pitch angle).
θ_0, $\hat{\theta}$	Initial value for the pitch angle and estimated pitch angle, respectively
θ_{ref}	Pitch angle calculated with the aid of data from air data and the slip sensors
$\bar{\varphi} = [\phi, \theta]^T$	Compressed symbol for roll angle and pitch angle
$\hat{\bar{\varphi}}$, $\hat{\bar{\varphi}}_0$, $\Delta\bar{\varphi}$	Respectively: estimated roll and pitch angle, estimated initial values for roll and pitch angle and difference between integrated-out and reference-calculated roll and pitch angle
φ_{ref}, φ_{AHRS}	Respectively: roll and pitch angle calculated with the aid of air data and primary data, and integrated-out roll and pitch angle, where integration is done with the aid of the angular rate gyro signals
ψ, ψ^i	Respectively: angle between the projection of x_B in the horizontal plane and north (heading angle), and discrete indexing of heading angle
α	Angle between air-related rate vector projected on the z-axis in body frame and projected on the x-axis in body frame (angle of attack)
β	Angle between air-related velocity vector and air-related velocity vector projected on the y-axis in body frame (sideslip angle)
C_B^N	Transformation matrix (3 x 3 matrix) which transforms a vector from body frame (actual) to navigation frame. The elements of this matrix are designated c_{11}, c_{12}, c_{13}, c_{21}, c_{22}, c_{23}, c_{31}, c_{32}, c_{33}, where the index designates row and column in that order
$C_B^N \cdot C_{\hat{B}}^B = C_{\hat{B}}^N = \hat{C}_B^N$	Transformation matrix which transforms a vector from body frame (calculated) to navigation frame

Table 1 Explanation of designations (symbols) for angles and angular rates. See also Figure 4.

δC_B^N	Difference between calculated and true C_B^N
$\gamma = (\gamma_x, \gamma_y, \gamma_z)^T$	Rotation around, respectively, the x-, y- and z-axis in body frame, corresponding to the error between true and calculated body frame
Γ	Anti-symmetrical matrix form of the vector γ
$\omega_{IB} = \omega = (\omega_x, \omega_y, \omega_z)^T$	Angular rate around, respectively, the x-, y- and z-axis in body frame (angular rate gyro signals). These angular rate components are customarily also designated $(p, q, r)^T$
W_{IB}	The vector ω_{IB} expressed in anti-symmetrical matrix form
$\delta\omega = (\delta\omega_x, \delta\omega_y, \delta\omega_z)^T$	Difference between actual and measured angular rate around, respectively, the x-, y- and z-axis in body frame
W_{IN}	Rotation of navigation frame relative to inertial frame that occurs when moving over the curved surface of the earth. Anti-symmetrical matrix form

Table 2 Explanation of symbols for the geomagnetic field.

B_x, B_y, B_z	Geomagnetic field vector components in body frame
$\delta B_x, \delta B_y, \delta B_z$	Difference between measured and actual field vector components in body frame
B_N, B_B	Geomagnetic field vector in navigation frame and body frame, respectively

Table 3 Explanation of symbols used in connection with filters

$k, k+1$	Used as an index, and represent the instant before and after time updating, respectively
$n, n+1$	Used to represent the present and subsequent sample, respectively
$-, +$	Used as an index, and represent the instant before and after measurement updating, respectively
x, z, P	State vector, measurement vector and estimate uncertainty matrix
w, Q	Process noise vector and covariance matrix for process noise, respectively

Table 3 Explanation of symbols used in connection with filters

A, F	Prediction matrix in time-continuous and time-discrete form
K, H, R	Kalman gain matrix, measurement matrix and covariance matrix for measurement noise, respectively
u_ω, u_b, u_s	Driving noise for the Markov processes
τ_ω, τ_b, τ_s, τ, τ_1, τ_2	Time-constants
f_s	Sampling frequency

Table 4 Explanation of other symbols. See also Figure 5.

b_x, b_y, b_z	Bias (zero errors)
s_x, s_y, s_z	Scale factor errors
k_{xy}, k_{xz}, k_{yx}, k_{yz}, k_{zx}, k_{zy}	Cross-connection errors (for example, index xy stands for how the y-component affects the x-component). Arise because the axes in triad are not truly orthogonal.
h, $\dot{h}$	Altitude and low-pass-filtered time-derived altitude respectively
v_t, v	True speed relative to the air
g	Gravity

CLAIMS

1. Method for synthetically calculating redundant attitude for an aircraft when the heading of the aircraft is known, with the aid of data existing in the aircraft, such as the angular rates p, q, r around the x-, y- and z- coordinates of an aircraft-fixed (body frame) coordinate system, air data information in the form of speed, altitude and angle of attack as well as heading information, **characterised in that** the method includes the steps:
- attitude is calculated on the basis of the aircraft-fixed angular rates p, q, r and
- the calculated attitude is corrected by means of air data and heading.

2. Method according to claim 1, **characterised in that** the heading information is obtained from a heading gyro.

3. Method according to claim 1 or 2, **characterised in that** attitude is integrated out via information about the body-frame angular rates (p, q and r) obtained from the aircraft-fixed angular rate gyros of the aircraft.

4. Method according to claim 3, **characterised in that** correction of the integrated-out attitude takes place with the aid of attitude calculated on the basis of air data information and heading information.

5. Method for synthetically calculating redundant attitude and redundant heading for an aircraft with the aid of data existing in the aircraft, such as the angular rates p, q, r around the x-, y- and z- coordinates of an aircraft-fixed (body frame) coordinate system, air data information in the form of speed, altitude and angle of attack, **characterised in that** the method includes the steps:
- attitude and heading are calculated on the basis of the body-frame angular rates p, q, r
- the errors in the measured body-frame magnetic field vector components are estimated,
- the measured body-frame field magnetic field vector is derived,
- errors in calculated attitude and heading are estimated with the aid of air data and derived measured body-frame magnetic field vector components and
- the calculated attitude and heading are corrected by means of estimated errors in attitude and heading.

6. Method according to claim 5, **characterised in that** attitude andr heading are integrated out via information about the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros.

7. Method according to claim 5, **characterised in that** estimation of errors in measured body-frame magnetic field vector components is performed in a first filter (11).

8. Method according to claims 6 or 7, **characterised in that** in a second filter (22) is performed estimation of attitude errors and heading errors that arise on integration of the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros, where the estimation is done with the aid of attitude calculated from air data information as well as derived measured body-frame magnetic field vector components.

9. Method according to claims 7 or 8, **characterised in that** the filtering takes place with the aid of Kalman filters.

10. Arrangement for synthetically calculating redundant attitude for an aircraft when the aircraft's heading is known, with the aid of data existing in the aircraft such as the aircraft's body-frame angular rates (p, q and r), air data including at least speed, altitude and angle of attack as well as heading information, **characterised in that** the arrangement includes an integration routine (8) to integrate out the aircraft's attitude from information about the aircraft's body-frame angular rates (p, q and r) as well as that the calculated attitude is corrected by means of reference attitude from air data and redundant heading.

11. Arrangement according to claim 10, **characterised in that** the heading information is obtained from a heading gyro.

12. Arrangement according to claim 10 or 11, **characterised in that** integration routine (8) integrates out the aircraft's attitude from the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros.

13. Arrangement according to claim 12, **characterised in that** the integration routine (8) is fed with the zero-error-compensated body-frame angular rate gyro signals.

14. Arrangement according to claim 10, **characterised in that** a reference attitude is calculated with air data information as well as redundant heading information.

15. Arrangement according to claim 10, **characterised in that** a synthetically-generated corrected attitude is obtained by generating a difference between the attitude obtained from the integration routine (8) and an error signal that represents the error between the integrated attitude and the reference attitude.

16. Arrangement for synthetically calculating redundant attitude and redundant heading for an aircraft with the aid of data existing in the aircraft such as measured body-frame field vector components, the aircraft's body-frame angular rates (p, q and r) as well as air data including at least speed, altitude and angle of attack, **characterised in that** the arrangement includes a first measurement routine (10) which transforms the measured body-frame magnetic field vector components to the aircraft's navigation system (navigation frame), a first filter (11) which estimates the errors in the calculated measured body-frame field vector components, an integration routine (20) for integrating out the aircraft's attitude and heading from information about the aircraft's body-frame angular rates (p, q and r), a second filter (22) for estimating the errors arising in attitude and heading obtained in the said integration and a second measurement routine (21) for calculating attitude and heading from air data and derived measured body-frame magnetic field vector components.

17. Arrangement according to claim 16, **characterised in that** the first measurement routine (10) is fed with the measured body-frame magnetic field vector components, as well as attitude and heading from the aircraft's normal navigation system and transforms the measured body-frame magnetic field vector components to the aircraft's navigation frame.

18. Arrangement according to claim 17, **characterised in that** the first filter (11) is fed with information from the first measurement routine (10) and estimates the errors in the measured body-frame magnetic field vector components.

19. Arrangement according to claim 16, **characterised in that** the integration routine (20) integrates out the aircraft's attitude and heading from the aircraft's body-frame angular rates (p, q and r) obtained from the aircraft's body-frame angular rate gyros.

20. Arrangement according to claim 16, **characterised in that** the second measurement routine (21) is fed with air data, the derived measured body-frame magnetic field vector components and with information about the aircraft's body-frame angular rates (p, q and r) and from these values calculates an attitude and a heading.

21. Arrangement according to claim 20, **characterised in that** the second filter (22) is fed with information from the second measurement routine (21) and estimates the errors in attitude and heading as well as zero error in body-frame angular rate gyro signals and residual errors in the measured body-frame magnetic field vector components for generating an error signal.

22. Arrangement according to claim 21, **characterised in that** a synthetically-generated corrected attitude and heading are obtained by generating a difference between
- the attitude obtained from the integration routine (20) and heading and
- the error signal from the second filter (22).

23. Arrangement according to claim 19, **characterised in that** the integration routine (20) is fed with body-frame angular rate gyro signals compensated for estimated zero errors.

24. Arrangement according to any of claims 16 - 23, **characterised in that** the first filter (11) and/or the second filter (22) consists of a Kalman filter.

1/3

Fig. 1

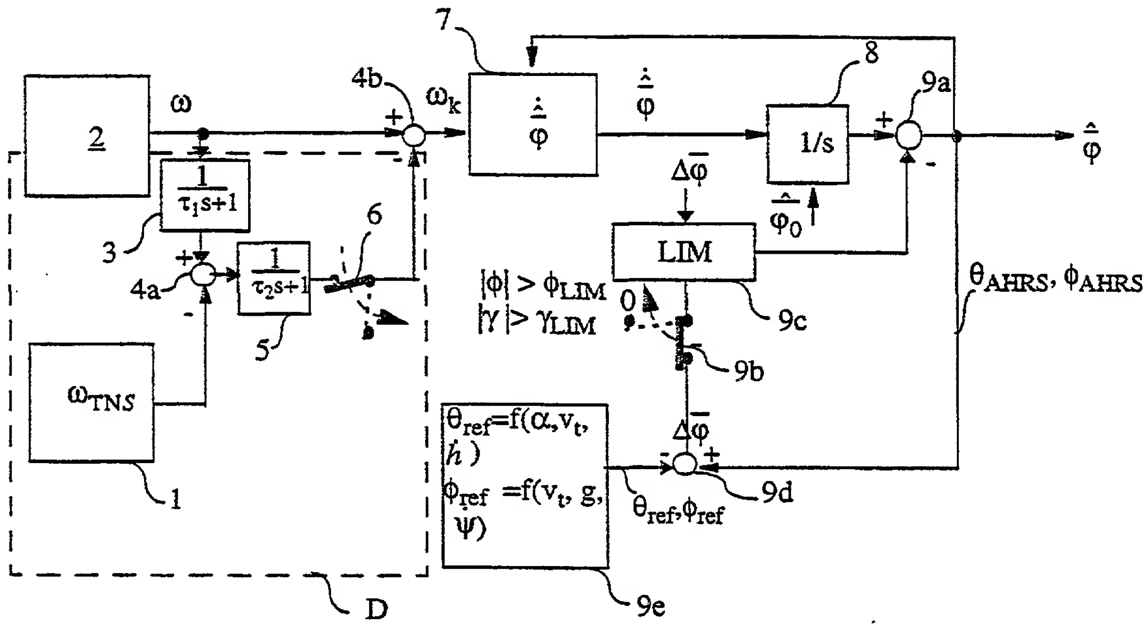
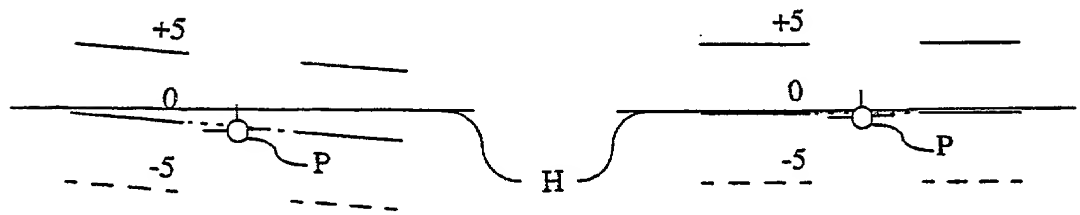

2
ω
4b
+
ω_k
7
$\dot{\hat{\varphi}}$
$\dot{\hat{\varphi}}$
8
1/s
9a
+
-
$\hat{\varphi}$
$\frac{1}{\tau_1 s+1}$
3
4a
+
-
$\frac{1}{\tau_2 s+1}$
6
5
$\Delta\bar{\varphi}$
$\hat{\varphi}_0$
LIM
$|\phi| > \phi_{LIM}$
$|\gamma| > \gamma_{LIM}$
0
9c
9b
$\theta_{AHRS}, \phi_{AHRS}$
ω_{TNS}
1
$\theta_{ref}=f(\alpha, v_t, \dot{h})$
$\phi_{ref} = f(v_t, g, \psi)$
$\Delta\bar{\varphi}$
-
+
9d
θ_{ref}, ϕ_{ref}
D
9e

Fig. 2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0
-5
P
H
+5
0
-5
P

2/3

Fig. 3

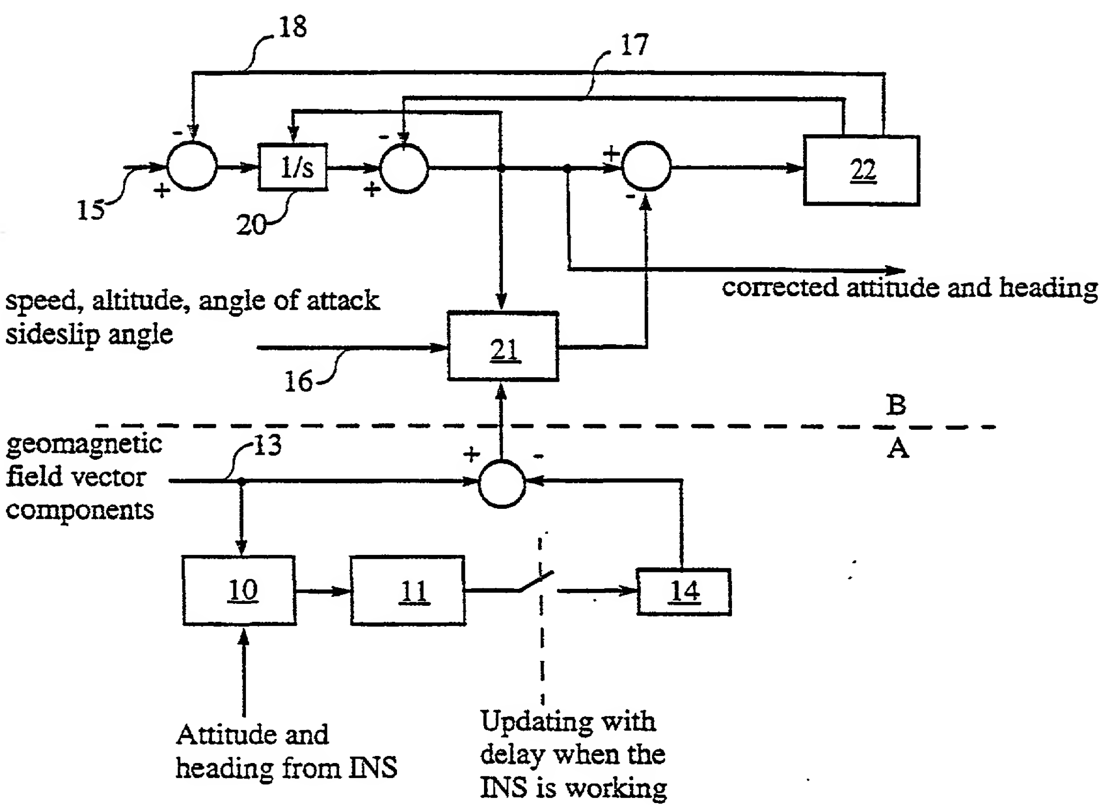

18
17
1/s
22
15
20
speed, altitude, angle of attack
sideslip angle
21
corrected attitude and heading
16
B
A
geomagnetic
field vector
components
13
10
11
14
Attitude and
heading from INS
Updating with
delay when the
INS is working

Fig. 4

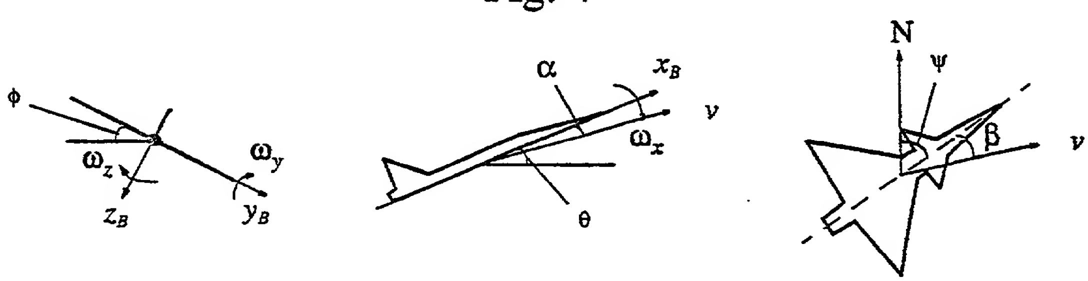
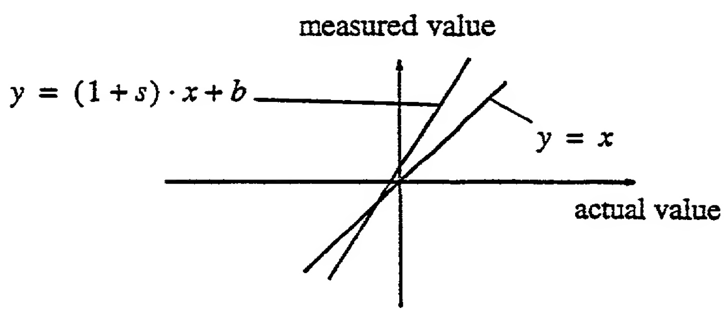

N
ψ
β
v
α
θ
ϕ

Fig. 5



890 US

Docket No. 19391.0025

COMBINED DECLARATION AND POWER OF ATTORNEY FOR ORIGINAL, DESIGN, NATIONAL STAGE OF PCT, SUPPLEMENTAL, DIVISIONAL, CONTINUATION OR CONTINUATION-IN-PART APPLICATION

As a below named inventor, I hereby declare that:

My residence, post office address and citizenship are as stated below next to my name,

I believe I am the original, first and sole inventor (if only one name is listed below) or an original, first and joint inventor (if plural names are listed below) of the subject matter which is claimed and for which a patent is sought on the invention entitled:

REDUNDANT SYSTEM FOR THE INDICATION OF HEADING AND ATTITUDE IN AN AIRCRAFT

the specification of which

a. ☒ is attached hereto

b. ☐ was filed on _____ as application Serial No. _____ and was amended on _____. (if applicable).

PCT FILED APPLICATION ENTERING NATIONAL STAGE

c. ☒ was described and claimed in International Application No. PCT/SE00/00034 filed on 12 January 2000 and as amended on _____. (if any).

I hereby state that I have reviewed and understand the contents of the above-identified specification, including the claims, as amended by any amendment referred to above.

I acknowledge the duty to disclose information which is material to patentability as defined in 37 C.F.R. § 1.56.

I hereby specify the following as the correspondence address to which all communications about this application are to be directed:

SEND CORRESPONDENCE TO:

Edward A. Pennington
SWIDLER BERLIN SHEREFF FRIEDMAN, LLP
3000 K Street, N.W., Suite 300
Washington, D.C. 20007-5166

DIRECT TELEPHONE CALLS TO: Eric J. Franklin, 202-424-7500

☒ I hereby claim foreign priority benefits under Title 35, United States Code § 119 (a)-(d) or under § 365(b) of any foreign application(s) for patent or inventor's certificate or under § 365(a) of any PCT international application(s) designating at least one country other than the U.S. listed below and also have identified below such foreign application(s) for patent or inventor's certificate or such PCT international application(s) filed by me on the same subject matter having a filing date within twelve (12) months before that of the application on which priority is claimed:

☒ The attached 35 U.S.C. § 119 claim for priority for the application(s) listed below forms a part of this declaration.

Country/PCT	Application Number	Date of filing (day, month, yr)	Date of issue (day, month, yr)	Priority Claimed
Sweden	9900113-3	18 January 1999		☒ Y ☐ N
				☐ Y ☐ N
				☐ Y ☐ N

☐ I hereby claim the benefit under 35 U.S.C. § 119(e) of any U.S. provisional application(s) listed below.

Provisional Application No.	Date of filing (day, month, yr)
____	____

ADDITIONAL STATEMENTS FOR DIVISIONAL, CONTINUATION OR CONTINUATION-IN-PART OR PCT INTERNATIONAL APPLICATION(S DESIGNATING THE U.S.)

I hereby claim the benefit under Title 35, United States Code § 120 of any United States application(s) or under § 365(c) of any PCT international application(s) designating the U.S. listed below.

US/PCT Application Serial No.	Filing Date,	Status (patented, pending, abandoned)/ U.S. application no. assigned (For PCT)
US/PCT Application Serial No.	Filing Date,	Status (patented, pending, abandoned)/ U.S. application no. assigned (For PCT)

☐ In this continuation-in-part application, insofar as the subject matter of any of the claims of this application is not disclosed in the above listed prior United States or PCT international application(s) in the manner provided by the first paragraph of Title 35, United States Code, § 112, I acknowledge the duty to disclose material information as defined in Title 37, Code of Federal Regulations, § 1.56(a) which occurred between the filing date of the prior application(s) and the national or PCT international filing date of this application.

I hereby declare that all statements made herein of my own knowledge are true and that all statements made on information and belief are believed to be true; and further that these statements were made with the knowledge that willful false statements and the like so made are punishable by fine or Imprisonment, or both, under Section 1001 of Title 18 of the United States Code and that such willful false statements may jeopardize the validity of the application or any patent issued thereon.

I hereby appoint the following attorneys and/or agents with full power of substitution and revocation, to prosecute this application, to receive the patent, and to transact all business in the Patent and Trademark Office connected therewith: Edward A. Pennington (Reg. No. 32,588), John P. Moran (Reg. No. 30,906), Eric J. Franklin (Reg. No. 37,134), Michael A. Schwartz (Reg. No. 40,161), Robert C. Bertin (Reg. No. 41,488), Alicia A. Meros (Reg. No. 44,937), Chadwick A. Jackson (Reg. No. 46,495), Edward J. Naidich (Reg. No. 43,826), and Sean O'Hanlon (Reg. No. 47,252) of Swidler Berlin Shereff Friedman having an address of 3000 K Street, N.W., Suite 300, Washington, D.C. 20007-5116.

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1-PD

Full name of sole or first inventor Peter Adebjörk

X

Inventor's signature Peter Adebjörk

Date: Aug 2, 2001

Residence: Gryningsgatan 73, S-589 29 Linköping, Sweden SEX

Citizenship: Sweden

Post Office Address: Gryningsgatan 73, S-589 29 Linköping, Sweden

2-PD

Full name of second inventor Per-Johan Nordlund

Inventor's signature Per-Johan Nordlund

X

Date: Aug 14, 2001

Residence: Vretagatan 17, S-582 37 Linköping, Sweden SEX

Citizenship: Sweden

Post Office Address: Vretagatan 17, S-582 37 Linköping, Sweden

3-PD

Full name of third inventor Carl-Olof Carlsson

X

Inventor's signature Carl-Olof Carlsson

Date: Aug 2, 2001

Residence: Skogslyckegatan 11D, S-587 27 Linköping, Sweden SEX

Citizenship: Sweden

Post Office Address: Skogslyckegatan 11D, S-587 27 Linköping, Sweden